REPORT DOCUMENTATION PAGE

| 1. REPORT DATE (DD-MM-YYYY)
22-06-2007 | 2. REPORT TYPE
Technical Paper | 3. DATES COVERED (From - To) |
|---|---|---|

4. TITLE AND SUBTITLE

Microarcjet Microthruster for Nanosat Applications (Preprint)

5a. CONTRACT NUMBER
FA8650-04-C-2514

5b. GRANT NUMBER

5c. PROGRAM ELEMENT NUMBER

6. AUTHOR(S)
John Slough (University of Washington); J.J. Ewing (Ewing Technology Associates)

5d. PROJECT NUMBER
OSDBR4QL

5e. TASK NUMBER

5f. WORK UNIT NUMBER

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

Ewing Technology Associates
5416 143rd Avenue SE
Bellevue WA 98006

8. PERFORMING ORGANIZATION REPORT NUMBER

AFRL-PR-ED-TP-2007-339

9. SPONSORING / MONITORING AGENCY NAME(S) AND ADDRESS(ES)

Air Force Research Laboratory (AFMC)
AFRL/PRS
5 Pollux Drive
Edwards AFB CA 93524-70448

10. SPONSOR/MONITOR'S ACRONYM(S)

11. SPONSOR/MONITOR'S NUMBER(S)
AFRL-PR-ED-TP-2007-339

12. DISTRIBUTION / AVAILABILITY STATEMENT

Approved for public release; distribution unlimited (07248A)

13. SUPPLEMENTARY NOTES
For presentation at the 43rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Cincinnati, OH, 8-11 July 2007.

14. ABSTRACT

This paper summarizes experiments and analysis of the micro-discharge microarcjet thruster. A small, very low power (2-5 Watt) micro-discharge was maintained between two electrodes in geometries compatible with application to a cold gas thruster. To evaluate the efficiency of the discharge in providing an increase in Isp and thrust, a special torsional micro-balance thrust stand, capable of micro-Newton resolution, was designed and constructed. The micro-balance thrust stand was installed in a large dielectric chamber with high pumping speed to eliminate stray coupling of the discharge with the vacuum chamber or background gas. A battery operated driver provided the 240-280 volts required to sustain the very low current (5-40 mA) arc discharge. The discharge was also ballasted inductively to avoid capacitive effects as well as resistive losses. Thrust measurements using a variety of electrode geometries and propellants were carried out. It was found that the optimum scale for the discharge bore for these low power levels was ~ 300 micron. Smaller bores resulted in too large a power loss in thermal transfer to the thruster/nozzle body. A larger bore led to large mass flows at the pressures required to produce a stable discharge. All ring or cavity electrode structures showed no appreciable gain and were rapidly eroded by the discharge. The only electrode configuration to show an increase in thrust from the application of the discharge with no measurable electrode erosion was in a configuration similar to the standard arcjet. Most of the studies were conducted in Argon, but other gases from H to Xe were also used. In Argon the optimal mass flow was ~ 2 mg/s with between 10 to 20 kPa upstream of the nozzle.

15. SUBJECT TERMS

| 16. SECURITY CLASSIFICATION OF: | | | 17. LIMITATION OF ABSTRACT | 18. NUMBER OF PAGES | 19a. NAME OF RESPONSIBLE PERSON
Dr. Jean-Luc Cambier |
|---|---|---|---|---|---|
| a. REPORT
Unclassified | b. ABSTRACT
Unclassified | c. THIS PAGE
Unclassified | SAR | 8 | 19b. TELEPHONE NUMBER
(include area code)
N/A |

Microarcjet Microthruster for Nanosat Applications (Preprint)

John Slough[1]
University of Washington, Seattle, WA, 98195

J.J Ewing[2]
Ewing Technology Associates, Bellevue, WA, 98006

This paper summarizes experiments and analysis of the micro-discharge microarcjet thruster. A small, very low power (2-5 Watt) micro-discharge was maintained between two electrodes in geometries compatible with application to a cold gas thruster. To evaluate the efficiency of the discharge in providing an increase in Isp and thrust, a special torsional micro-balance thrust stand, capable of micro-Newton resolution, was designed and constructed. The micro-balance thrust stand was installed in a large dielectric chamber with high pumping speed to eliminate stray coupling of the discharge with the vacuum chamber or background gas. A battery operated driver provided the 240-280 volts required to sustain the very low current (5-40 mA) arc discharge. The discharge was also ballasted inductively to avoid capacitive effects as well as resistive losses. Thrust measurements using a variety of electrode geometries and propellants were carried out. It was found that the optimum scale for the discharge bore for these low power levels was ~ 300 micron. Smaller bores resulted in too large a power loss in thermal transfer to the thruster/nozzle body. A larger bore led to large mass flows at the pressures required to produce a stable discharge. All ring or cavity electrode structures showed no appreciable gain and were rapidly eroded by the discharge. The only electrode configuration to show an increase in thrust from the application of the discharge with no measurable electrode erosion was in a configuration similar to the standard arcjet. Most of the studies were conducted in Argon, but other gases from H to Xe were also used. In Argon the optimal mass flow was ~ 2 mg/s with between 10 to 20 kPa upstream of the nozzle. With 5 W of electrical power into the discharge the thrust was increased 1.1 to 1.8 mN with an effective Isp of 90 s. The highest efficiencies (~ 40%) were obtained at lower power indicating enhanced thermal losses at higher Isp (gas temperature).

Nomenclature

A_{moly}, A_{insl}	=	outer surface area of molybdenum nozzle, and thruster body insulators
ε_{moly}, ε_{insl}	=	emissivity of molybdenum, and insulators
Isp	=	specific impulse (s)
$P_{I \cdot V}$	=	electrical input power to thruster
P_{jet}	=	thruster jet power
σ	=	Boltzmann constant
T	=	thrust
T_b	=	bulk temperature of thruster body

I.   Introduction

WITH the development of small (kg) sized spacecraft, there will be a need for micropropulsion devices that are not only very low mass but can also operate at very low power. In response to this need work on a wide array of devices has taken place[1-3]. Future nano-spacecraft (nanosats) will require significant propulsion capability in order to provide a high degree of maneuverability and capability. Given the likely presence of onboard electrical power generation, there would be significant benefit from using electric propulsion (EP) for improving the efficiency of propellant use as well as an increase in exhaust velocity (Isp). The leveraged use of spacecraft assets in

---

[1]Department of Aeronautics and Astronautics, Box 352400, AIAA Member
[2]

American Institute of Aeronautics and Astronautics

this way adds to the simplicity of operation and reduction of spacecraft mass. Current efforts under development are targeted for spacecraft mass ~100 kg with an available power level for propulsion of ~100 watts. There are several propulsion systems, initially developed for much higher powers that are now being redesigned for this lower power regime. In this group, significant development has been put into low power versions of ion thrusters, Hall thrusters, and the PPT[4]. There are also novel devices that work primarily at low power such as field emission thrusters, vaporizing liquid thrusters, resistojets, microwave arcjets, and micro-arcjets[5].

The available power for propulsion is typically on the order of a watt per kilogram. These devices would have applicability at the 100 watt level and above, and would be suitable for microsat applications. A nanosat (10 kg and lower) requires a thruster that is simple and requires very little power processing and related hardware. A cold gas thruster fed by a micro valve and propellant tank is an example of the kind of simplicity that is desired[6]. Assuming that a thruster of this type is already available onboard the nanosat, it would be advantageous to add additional capability to this device with the inclusion of the available onboard power which would most likely be in the range of 1 to 10W at low voltage. There have been only limited investigations of EP thrusters that operate at 10 watts and below. A first choice would be to add energy to the gas via a DC discharge. The simplest geometry that one can imagine for a micro-thruster in the context of a very low power DC discharge is that shown in Fig. 1. In this scheme, one has two electrodes separated by an insulator with a small hole bored through. The insulator acts as a discharge gap as well as nozzle in this configuration. This electrode configuration was investigated previously[7]. It was found in this study that there was very little thrust enhancement from this configuration. It has been demonstrated that a very low power DC arcjet can provide for a fairly efficient thruster (up to 40%) in the 10 to 40 watt range at an Isp of 200 to 300s[8]. The structural simplicity of an arcjet is favorable for both size and mass reduction of the conventional thruster. The objective of this study is to investigate the fundamental discharge characteristics and performance of this type of very low power DC discharge. Specifically, the aim was to study the discharge in a range of electrical input power from approximately 1 to 5 watts, and in a configuration suitable for generating thrust in a vacuum. The next section discusses the design of the devices tested. A thrust stand for measuring the micro-arcjet microthruster and other configurations was constructed to determine which of the possible electrode configurations provided for the best performance in terms of thrust, Isp, and efficiency.
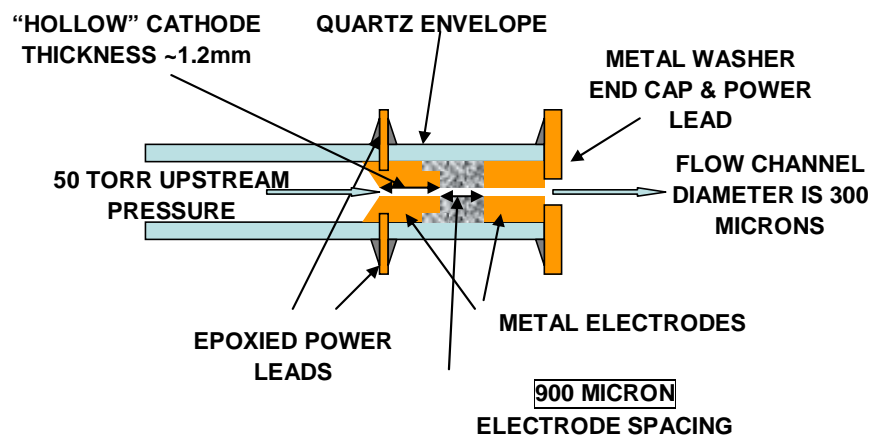

Figure 1 shows a typical structure using the electrodes on the outer wall of the flow channel. *These types of structures showed little discharge enhancement to the cold gas thrust.*

American Institute of Aeronautics and Astronautics

II.   Thruster Design

A series of designs that consisted of variants on the flow-through topology were tested. Like the design shown in Fig. 1, all had in common discharge electrodes that were at the edge of the cylindrical flow channel. For typical flow rates of 1mg/sec, the observed cold gas thrust was in the range of 0.5 to 1mN and discharge enhancement of the cold gas thrust from the discharge was negligible at all power levels tested (1-10 W). Any heating of the gas stream was lost to the thruster wall prior to the exit nozzle. It was quite common to find the discharge being maintained even further upstream of the upstream electrode. Cathode erosion was always severe limiting thruster operation to several hours at most.

The only electrode configuration to show an increase in thrust from the application of the discharge was in an arrangement very similar to the standard arc jet. The upstream cylindrical electrode was a tungsten wire with a diameter of 300 microns. The downstream electrode was a molybdenum nozzle with a 45 degree cone angle, and a bore of 350 microns at the constricted end. The tungsten wire was held in place inside a 6 mm Pyrex® tube by a Macor® annular cylinder. The most stable discharge was maintained by having the tungsten electrode biased as the anode as opposed to the usual cathode arrangement in the standard arcjet. When the micro-arcjet was operated in the standard manner, it was difficult to keep the discharge from propagating upstr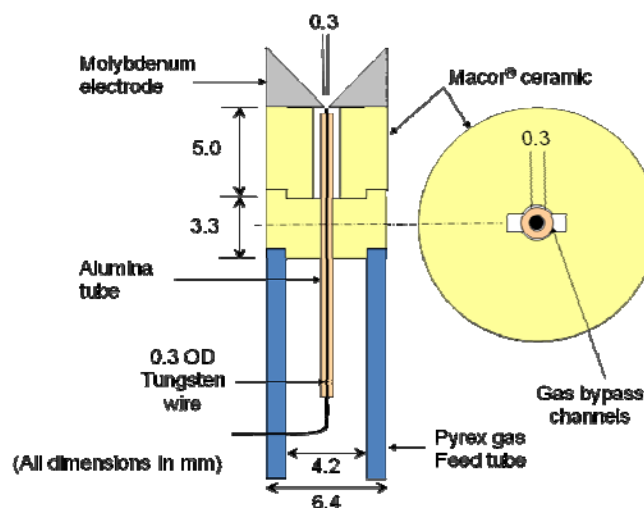eam into the gas feed line. When this occurred, the thrust enhancement was dramatically reduced to essentially that obtained with the cold gas alone. All other electrode configurations, including the myriad of annular electrode structures suffered from this same behavior.

Figure 2. Micro-arcjet thruster. This version is the design that was used for the measurements reported here.

Basically, any discharge occurring in the gas flow prior to the exit nozzle, even when between the electrodes, gave negligible to no contribution to the cold gas thrust. It was possible to operate in the standard discharge mode with the cathode in the upstream by making sure that the tungsten wire was well insulated from the propellant gas from the tip to several cm upstream where the wire leaves the thruster body. This was provided by as close-fitting alumina tube around the cathode wire. The final design used in these studies is shown in Fig. 2.

It is interesting to note that the discharge characteristics are not that of a typical arc. The discharge voltage is always in the range of 200 to 300V rather than dropping below 50 V characteristic of an arcjet. This may be

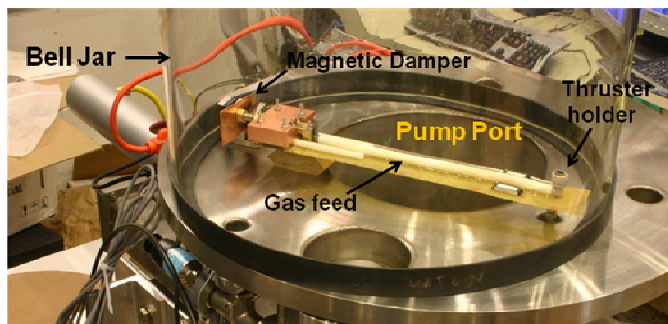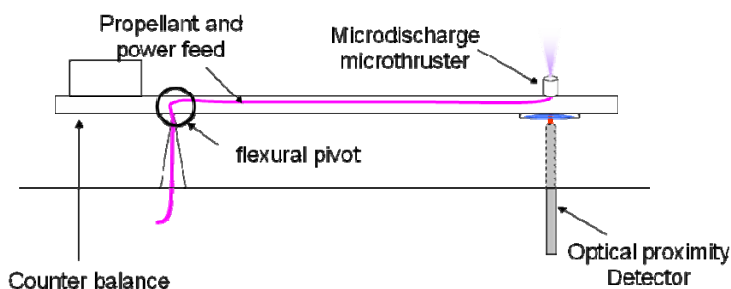**Figure 3. Micro-arjet thrust stand setup**. *(Top) schematic of showing key elements of the design. (Bottom) Picture of device in vacuum.*

American Institute of Aeronautics and Astronautics

due to the fact that the heating of the cathode is inadequate to raise the temperature of the cathode sufficiently to enhance the rate of thermionic emission.

III.   Thrust Stand for Micro-arcjet Thruster

In order to evaluate the efficiency of the discharge, a special torsional micro-balance thrust stand capable of micro-Newton resolution, was designed and constructed for the thrust measurements. The thrust stand was installed in a dielectric chamber with high pumping speed (1,500 liters/sec). This was found to be necessary to eliminate stray coupling of the discharge with the vacuum chamber or background gas. The thrust stand was constructed primarily of dielectric materials to avoid electrical coupling to the discharge.

The thrust stand was basically a simple cantilever balance. The propellant and power leads were routed down the balance to the micro-thruster support mounted at the end of the arm. The support was located above an optical proximity fiber optic probe (Phlitec Model D163). The probe was positioned via a feedthru directly below the thruster mounting position on the FR-4 board that forms the backbone of the cantilever. This device records the vertical displacement of the thruster under thrust. A simple mirror is mounted on the bottom of the cantilever and forms the objective for the probe. The pivot uses torsion mounts to provide a known, though slight resistance to motion. The sensitivity of the position sensor with the torsional pivot employed (C-Flex model E20) at the fulcrum was 5 μN/mV. This was more than adequate as the thrust obtained from just the cold gas was in the milli-newton range with the micro-thrusters tested.

Because the thrust stand is essentially a balance, the calibration of the thrust stand was easily obtained. The characteristic deflection of the lever arm as a function of force (thrust) applied was obtained over the range of interest by applying a set of calibrated paper weights on the thruster mount and measuring the sensor voltage change. In this way any nonlinearity in the response of the entire system is taken into account in the calibration. With the use of a frictionless flexural pivot for tensioning there was little damping of the pivot arm motion. The period of the oscillation was ~ 1 sec and the oscillations would extend to several minutes. A magnetic damper was added at the counter balance end of the balance. This damper employed a copper sheet mounted between two poles of a samarium-cobalt permanent magnet pair. The sheet moves in the field, due to the undamped oscillation of the balance, and the multi-kG field of the permanent magnet pair sets up eddy currents in the copper sheet that dissipate the energy of the oscillation. Any motion is strongly damped in a few seconds, but the deflection due to a constant force is unaffected. The damper naturally limits the response time of the system to a few seconds, but this more than sufficient to measure the thrust and resolve changes on the thermal time of the thruster.

A battery operated power source (6 V DC) was constructed to provide both isolation and up to 600 volts required to initiate the discharge in what is believed to be a very low current (5-20 mA) arc discharge[7]. The discharge was also ballasted inductively (10 mH) to avoid capacitive effects as well as resistive losses. All gas and electrical connections to the thruster are brought along the lever arm. The wires and flex tubing are brought in at the pivot point along the axis of the pivot to minimize the torque on the pivot. The actual source and value of the torsional force is not important as long as it is in the range desired. As mentioned, any influence from these connections is accounted for in the calibration.

IV.   Results from Experiments

Thrust measurements using a variety of electrode geometries and propellants were carried out. It was found that the optimum scale for the discharge bore for these low power levels was ~ 300 - 400 micron. Smaller bores resulted in too large a power loss in thermal transfer to the thruster/nozzle body. A larger bore led to large mass flows at the pressures required to produce a stable discharge. Most of the studies were conducted in argon, but other gases from H to Xe were also used. In argon, the
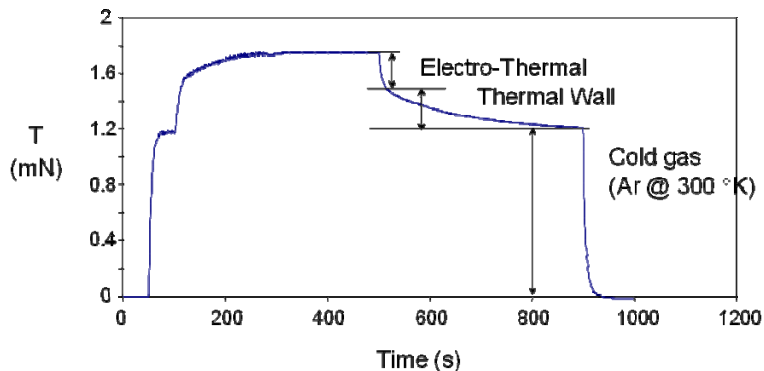

Figure 4. Thrust from micro-arcjet operating in Argon. *The input power was 2.25 W (8 mA @ 280V). The mass flow rate was 2 mg/sec.*

American Institute of Aeronautics and Astronautics

optimal mass flow was ~ 2 mg/s. The discharge voltage for sustainment was 230 – 280 V. Initiation of the discharge required the application of ~ 400 V indicating that the pressure was at or near the Paschen minimum. The mass flow was controlled by means of a mass flow controller, and the upstream pressure was monitored throughout the discharge by a pressure sensor. The initial cold gas upstream pressure was typically between 10 and 20 kPa. It increased 5 to 10 kPa during the discharge reflecting the increase in back pressure from the gas heating. The vacuum background pressure was constant at ~ 7 mPa and varied linearly with mass flow. The time history of the thrust for the micro-arcjet is shown in Fig. 4. The initial rapid increase is from the cold gas flow which in this case was 2 mg/sec Argon. After a period of several seconds to obtain a steady flow, the arc is initiated. There is a sudden rise in thrust (on the order of the response time of the thrust detection system and mass flow controller). This is clearly from the direct heating of the gas by the arc discharge. After this thrust increase there is a continued slower rise in thrust. It is believed that this rise is due to the indirect heating of the gas by thermal conduction due to the increased temperature of the thruster wall. This additional component is thus more like a resistojet rather than an arcjet. With the termination of the arc current, the thrust reverts back to the cold gas level. First there is a rapid drop as the direct heating of the gas from the arcjet disappears, and then there is the slow timescale reduction of the thrust as the thruster wall cools.
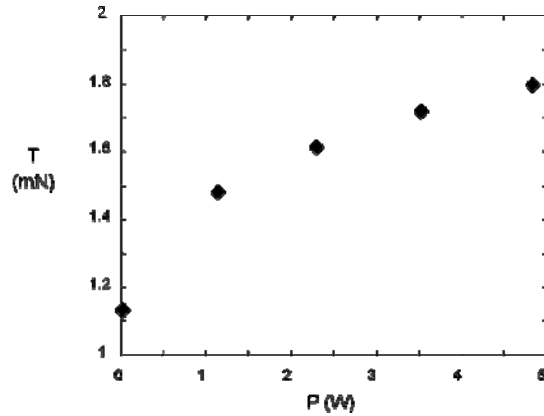

Figure 5. Final thrust as a function of input power into discharge. *The mass flow was 2 mg/sec Argon.*
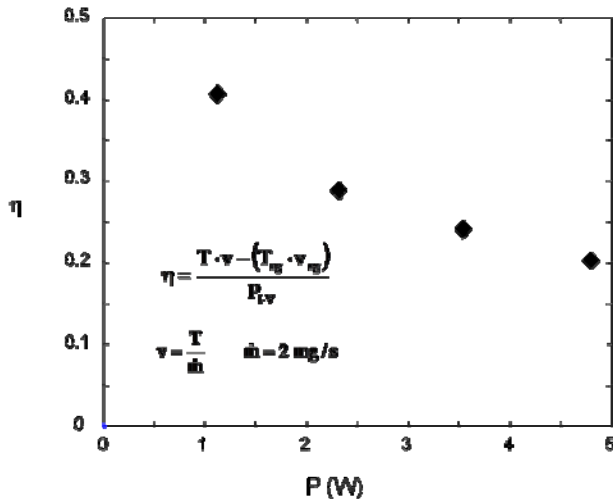

Figure 6. Thruster efficiency as a function of input power. *Data is based on formulas where the subscript cg refers to the cold gas measurements in Argon.*

The initial cold gas thrust in Argon was ~ 1.1 mN with an inferred Isp of 57 s. A plot of the thrust as a function of the input power is given in Fig. 5. The formulas used to calculate the various quantities of interest are included in Fig. 6. With 5 W of electrical power into the discharge the thrust was increased to 1.8 mN with an effective Isp of 90 s. There were minor variations in discharge voltage resulting from different electrode spacing and conditioning, but the resultant thrust was very similar at similar powers. Several different gasses were used which included hydrogen, helium, neon, nitrogen, argon, krypton, and xenon. There were variations in flow and voltage required for operation, but all performed with about the same thrust enhancement seen in Argon. The gas flow could be reduced, but this did not improve the efficiency observed from that of Argon (see Fig. 6). The highest efficiencies (~ 40%) were obtained at lower power indicating enhanced thermal losses at higher Isp (gas temperature). Improvements in Isp came only from employing a lower atomic or molecular mass propellant. The limiting factor was always the temperature. A thermocouple placed in contact with the inner nozzle surface indicated the same limiting temperature (~ 600 °K) regardless of gas species as it is the material properties of the thruster itself that matter here.

V. Discussion and Conclusions

The device shown in Fig. 2 on which this data was taken was quite reliable and exhibited no measurable erosion or mass loss after 100 continuous hours of operation at 6 watts. The bound on the mass loss was < 0.1mg which was the accuracy limit on the analytical balance used to weigh the thruster before installation and after demounting. The almost flat behavior of the I-V curve suggests that the core discharge is behaving like an arc but that the discharge in

American Institute of Aeronautics and Astronautics

the cathode fall region is behaving like a glow discharge. The long life of the electrodes even with the cathode downstream is entirely consistent with "glow like" performance in the cathode region.

As was mentioned, the temperature at the downstream nozzle electrode was measured during and after the discharge during a run where the thrust was brought to steady state. The steady state temperature of the nozzle was measured to be 570-580 °K regardless of propellant or propellant flow rate. This is consistent the following estimate of the electrode temperature based on the steady state power balance. The electrode temperature was obtained using a steady state approximation and effectively a one-dimensional thermal diffusion model. At steady state the power input to the electrode and supporting structure is balanced by power loss due to conduction, radiation and any transfer to the flow. The power input to the cathode is from ion bombardment in the cathode fall and heat transfer from the flowing gas. The power to the thruster from the flow is most likely negligible as the cooling rate after the discharge was the same regardless of whether gas continued to flow through the thruster, or was shut off.

The power input from the plasma component is due primarily from ion bombardment in the cathode fall, and electron thermal conduction at the thruster wall and anode. In any case, it is assumed that all of the power not lost as jet power shows up as heating of the thruster. For a mid range power input of 3W, it is assumed that ~ 2 W goes into the thruster.

The principal heat loss is by conduction and radiation. Thermal conduction can take place along 2 paths, into the Macor/Alumina insulators and then on to the Pyrex wall, as well as down the wires (see Fig. 2). The thermal conduction down the power lead wire is small due to the small diameter, as well as the long total length. It is estimated that less than ~ 40 mW of heat is conducted down the wires. The area of the insulator touching the electrode is much larger, ~ 30 mm^2, and as such can conduct a fair amount of heat. The thermal conductivity of the Macor/Alumina was taken to be ~ 10 W/m-K. In the initial stages of heating the electrodes the temperature drop across the insulators is on the order of 300°K and the Macor can conduct about as much heat as is being provided to the molybdenum electrode. The Macor will then heat to a temperature approximating that of the electrode. The heating stays localized to the thruster body due to the thermal insulation provided by the Pyrex tubing which has much lower conductivity. The area of the quartz or Pyrex tubing down which the heat can conduct is somewhat smaller, (~15 mm^2), and the thermal conductivity is much lower by a factor of 7. The steady state drain of power out of the thruster body is in the range of 0.4 W to 0.8 W at most. Since the thermal conduction depends on the magnitude of the thermal gradient, this will decrease substantially during the cool down. It is calculated that the time to equilibrate the nozzle electrode with the Macor is on the order of several hundred seconds based on the Macor volume and heat capacity and the estimated power flow. This is similar to the time required to reach full thrust or cool down lending support to the interpretation that the long term transient behavior into equilibrium is determined by thermal conduction and heating in the structure.

The bulk of the power loss must be in the form of radiation. Since the device is operated in vacuum, this is really the only mechanism that could be responsible. Radiation cooling is very strong function of temperature, and will effectively clamp the temperature. This is what was observed. If one analyzes the rate of the decay in the thruster temperature at turn off, it doesn't fit either a purely radiative decay or a conductive one. This is consistent with the view that radiation will dominate at high temperature and the conduction at lower temperature. The radiated power is given by the Stefan Boltzmann law, modified from the ideal by the emissivity of the various components. The steady state temperature is then given by the power balance equation, applied to the two different materials at a common bulk temperature. T_b:

$$(\epsilon_{moly} \, A_{moly} + \epsilon_{insl} \, A_{insl}) \, \sigma T_b^4 = P_{I \cdot V} - P_{jet}.$$

Molybdenum, being a metal, has an emissivity of order 10% at low temperature. Macor being an insulator has a higher emissivity which is taken to be 70%, the value of aluminum oxide at 500 °K. The areas for emission of each material is A_{moly} ~ 110 mm^2 and the outer area of the Macor, A_{insl} = 162 mm^2.

The difference in the input electrical power $P_{I \cdot V}$ and the jet power, P_{jet} was 2 W for the mid-range case. This power loss would require a bulk temperature T_b of ~ 600K, which is in reasonable agreement with the measurements. There can be significant error in several of the assumptions made here, but the strong temperature scaling of the radiative losses makes the final temperature insensitive to these assumptions.

While the efficiency of the thruster is not high, it is not that low, and compares favorably to the efficiency of other thrusters that might be employed on this small size and power scale. It is certainly simple, robust, and small. It also can be used as an addition to a cold gas thruster or operate as a stand alone unit. The inductive ballast and power source make for a very efficient method for actuating and maintaining the discharge[7] from the available power bus on a nanosat. It also provides for a stable discharge with minimal coupling to the spacecraft. A higher

range of Isp would be valuable for further decreasing the propellant requirements. It appears that this can only be obtained with lower atomic mass propellants using this thruster. These propellants usually come with tankage mass penalties. A further direction might be the improvement in thermal management. A different choice of materials or the isolation of the thruster walls and electrodes to limit conduction may allow for a higher gas temperature and efficiency. The observed temperatures are far from the limit for these materials so there is room for improvement.

Acknowledgments

References

[1]R.M Myers,.S.R. Oleson,, F.M. Curren, and S.J. Schneider, "Small Satellite Propulsion Options," AIAA Paper 94-2997, June 1994.

[2]J. Mueller, "Thruster Options for Microspacecraft: A Review and Evaluation of Existing Hardware and Emerging Technologies," AIAA Paper 97-3058, July 1997.

[3]M.M. Micci, and A.D. Ketsdever. Editors, "Micropropulsion for Small Spacecraft", Progress in Astronautics and Aeronautics Vol.187, American Institute of Aeronautics and Astronautics, 2000.

[4]M. Martinez-Sanchez and J.E. Pollard, "Spacecraft Electric Propulsion-An Overview", Journal of Propulsion and Power, Vol.14, No.5, 1998, pp.688-699.

[5]J.M. Sankovic and D.T. Jacobson, "Performance of a Miniaturized Arcjet", AIAA Paper 95-2822, July 1995.

[6]A.D. Ketsdever, D.C. Wadsworth, P.G. Wapner, M.S. Ivanov, G.N. Markelov, "Fabrication and Predicted Performance of Conical DeLaval Micronozzles" AIAA Paper 99-2724, July 1999.

[7]J. Slough, S. Andreason, T. Ziemba and J.J. Ewing, "*Micro-Discharge Micro-Thruster*", AIAA-2005-4074, 41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, Tucson, Arizona, July 10-13, 2005.

[8]H. Horisawa.and I. Kimura, "Studies of Very Low Power Arcjets", Chapter 6 in Micropropulsion for Small Spacecraft (M.M Micci and A.D. Ketsdever, eds.), Progress in Astronautics and Aeronautics Vol.187, pp.185-197, American Institute of Aeronautics and Astronautics, 2000.
.